BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: ATM Alternate Mission Study-
Performance Analysis - Case 610

DATE: September 3, 1968

FROM: I. Hirsch
K. E. Martersteck

ABSTRACT

An analysis has been made of the Saturn IB launch vehicle and spacecraft propulsion systems orbit-altitude capability for high inclination CM-SM/LM-ATM missions in either elliptical or circular orbits. This effort was part of an investigation of an alternate ATM mission flown independent of the Orbital Workshop in order to increase the fraction of solar viewing time by suitable optimization of the orbit parameters.

Altitude vs inclination data are presented for three different payload weights corresponding to spacecraft loaded for mission durations of 14, 28 and 56 days. A mission sequence was selected which allows flexibility in the optimization of the final orbit parameters, but still makes full utilization of both launch vehicle and spacecraft performance capability, including suborbital SPS ignition.

Instantaneous impact point plots were generated. By launching along an azimuth of 43.5° and applying a slow, constant yaw rate throughout the Sa-IB second-stage burn, the desired inclinations can be reached without jeopardizing the U. S. East Coast areas. It should be noted that several operational problems arise in conjunction with high-inclination missions such as significantly increased phasing time required for rendezvous and the potential of high-latitude landing and recovery.

N79-71580

(NASA-CR-73519) ATM ALTERNATE MISSION STUDY-PERFORMANCE ANALYSIS (Bellcomm, Inc.) 17 p

Unclas
00/12 11295

FF No. 602
CR-73519
(NASA CR OR TMX OR AD NUMBER) (CATEGORY)
[redacted]

BELLCOMM, INC.
1100 Seventeenth Street, N.W. Washington, D.C. 20036

SUBJECT: ATM Alternate Mission Study-Performance Analysis - Case 610

DATE: September 3, 1968

FROM: I. Hirsch
K. E. Martersteck

MEMORANDUM FOR FILE

Introduction

In the current baseline for mission AAP-3/AAP-4, the CM/SM and LM/ATM will dock to the AAP-2 Orbital Workshop in a clustered configuration. A number of contingencies may preclude this mission. In such an eventuality a different mission in which the solar observations are made with the LM/ATM docked directly to the CM/SM may be desirable. Such a mission, called the ATM alternate mission, has been investigated by a task force of Bellcomm personnel coordinated by G. M. Anderson of Department 1022. This memorandum reports the results of the rocket performance analysis performed by Department 1025 in support of ATM alternate mission study.

The basic motivation for the study was to increase the fraction of solar viewing time during the mission by increasing the inclination and apogee of the orbit. In the 28.5° inclination orbit of the Orbital Workshop, the ATM would be occulted about 42% of the time. Unocculted viewing can be achieved if the orbit plane is inclined enough to be perpendicular to the sun's rays. At the summer solstice when the sun's declination reaches 23.5°, an inclination of 66.5° would do. At lesser inclinations full-time viewing can still be achieved if the orbit is high enough to remain out of the earth's shadow. This is most economically achieved with an elliptical orbit oriented so that apogee is located on the "night side" of the earth away from the sun. Unfortunately, orbit perturbations and rotation of the earth-sun line cause the geometry to shift and the ideal conditions required for 100% solar viewing can be maintained only for a few days if at all. Thus, the choice of an orbit to maximize the cumulative viewing time is more complex than implied above and requires a parametric study with inclination, apogee height, solar declination, nodal position relative to the sun, and perigee location in the orbit plane as independent variables. This optimization is the subject of a companion paper (Reference 1) and will not be discussed here.

In order to optimally choose the free variables of season, node, and argument of perigee it is first necessary to know what values of apogee height and inclination can reasonably be achieved by the standard Saturn IB launch vehicle and standard spacecraft propulsion systems. This performance capability is the subject of this paper.

Mission Duration

Since spacecraft weight strongly influences performance, three different weights were considered corresponding to spacecraft loaded for mission durations of 14, 28, and 56 days. Although the 14-day mission was subsequently dropped from consideration in the overall study, some data developed for the 14-day spacecraft will be included here.

Orbit Lifetime

The first question to be answered was what orbits would provide adequate lifetime for these missions. Using the MSFC Orbital Lifetime Prediction Program (Reference 2), families of equal-lifetime ellipses (shown in Figure 1) were generated for the CM-SM/LM-ATM configuration flying in the ATM solar-oriented attitude. These data assume the $+2\sigma$ atmosphere density, i.e., -2σ lifetime, and a June 1971 launch. Rendezvous phasing considerations (discussed below) dictate a perigee of about 150 nm or greater for the mission sequence used for the missions. Therefore, for apogee altitudes of 200 nm or greater there should be no problem with orbital lifetime even for 56-day missions.

Launch Phase

To achieve the high inclinations of interest in this study, it is necessary to use a launch azimuth well beyond the approved Apollo range of 72° to 108°. Special attention must be paid to the track of the boost-phase instantaneous impact point (IIP) to ensure that it does not pass over populated land areas, except just before orbital insertion. (The IIP denotes where debris would land if thrust were suddenly and prematurely terminated. It should not be confused with the ground track of the vehicle.) High inclinations can only be achieved by launching to the north; southerly launches are precluded due to the easterly location of South America. The most northerly launch azimuth known to have been flown to date (unmanned Thor/Delta) is 43.5° which yields an orbital inclination of 50.6°. Steeper inclinations can be achieved with minimum jeopardy to the East Coast with a dog-leg trajectory. A launch azimuth is used that carries the IIP well out into the ocean. Then the thrust acceleration is deflected to the left by a yaw rotation, turning the velocity vector and IIP also to the left.

Simulations with first-stage yawing or a quick yaw turn at the beginning of the second-stage burn still produced IIP paths over populated portions of the East Coast (e.g., New York!). Better results were obtained with a slow and constant yaw rate throughout the second stage burn. Typical instantaneous impact point plots are shown in Figure 2 for two cases of interest: ~50° and ~63.5° inclinations, for a 28-day CM/SM flight using SPS suborbital ignition. The figure shows nominal-trajectory IIP's. A detailed dispersion analysis would produce a narrow band of possible IIP's on either side of the nominal IIP trace. In LM/ATM flights and two-stage-to-orbit CM/SM flights the S-IVB goes into orbit and does not impact during the launch phase.

It can be seen that the IIP for missions near 50° does not cross land until late in the suborbital powered flight. This case was studied rather extensively in conjunction with the former mission AAP-1A and could probably win range safety approval. As the inclination is increased, the IIP's pass over Newfoundland and Nova Scotia. Although the IIP passes over these areas relatively quickly (about 16 seconds over Newfoundland for the 63.44° case in Figure 2), this would present a problem for the range safety approval.

On-Orbit Phase

The sequence of maneuvers after the spacecraft reach orbit must allow flexibility in the selection of the final orbit parameters for solar viewing optimization, but also capitalize as much as possible on both launch vehicle and spacecraft performance capability. The sequence depicted in Figure 3 does satisfy these requirements and was selected for consideration in this study.

First the LM/ATM is launched into an 81 x 150 nm orbit. Using its own RCS system, the LM circularizes its orbit at 150 x 150 nm by an apogee burn and awaits the arrival of the CM/SM. The CM/SM is launched into the usual 81 x 120 nm orbit, performs the requisite phasing maneuvers and rendezvous with the LM/ATM in the 150 nm circular orbit. The docked spacecraft combination is then inserted into the operational elliptical orbit with perigee at 150 nm by burning the SM SPS engine. After the ATM observations are completed, the CM/SM undocks from the LM/ATM and using the SPS at perigee again circularizes the orbit at 150 nm. Finally the deorbit maneuver is made with the SPS engine. In case of failure of the SPS, sufficient RCS propellant would be carried for a backup deorbit burn at apogee.

The elliptical insertion allows a greater LM/ATM weight to be orbited than with the more-conventional circular insertion. It is felt that the 150 nm parking-orbit altitude is the minimum which would permit reasonable rendezvous phasing for the CM/SM. Use of the circular parking orbit is a propulsively efficient way to provide complete freedom in the position of apogee. Similarly, circularization before the final deorbit maneuver allows greater flexibility in the selection of a landing site.

Circular orbits at the high inclinations were also investigated. For these cases the SM SPS is used to perform a Hohmann transfer from the 150 nm parking orbit to the final circular operational orbit. Both primary (SPS) and backup (RCS) deorbit maneuvers would be made directly from the operational orbit.

Simulation Results

Using a specially modified version of the Bellcomm Apollo Simulation Program described in Reference 3, the data shown in Figures 4 and 5 were generated for several missions of the type described above. Appropriate spacecraft weights for the missions listed below were derived by W. W. Hough (Reference 4):

	Spacecraft (no usable propellant)	
Duration	CM/SM	LM/ATM
14 days	24760 lbs	25694 lbs
28 days	26070 lbs	25834 lbs
56 days	28234 lbs	26114 lbs

RCS propellant was added to the LM as follows: 400 lbs for orbit circularization and 50 lbs for attitude control. RCS propellant was added to the SM as required for orbit maneuvers and cluster control, rendezvous and backup deorbit; SPS propellant for orbit insertion where suborbital ignition was used, for orbit transfer maneuvers and for primary deorbit was also included. Standard AAP weights for launch vehicle stages, SLA, nose cone, etc., were used.

The simulator was programmed so that all SPS propellant which could be orbited in the SM and not needed for phasing or deorbit was used to transfer the CM-SM/LM-ATM to as high an energy orbit as possible (either with perigee of 150 nm or circular). 1000 lbs of SPS propellant in the SM were included for flight performance reserve (FPR) when SPS suborbital ignition was used. In two-stage-to-orbit cases, 1500 lbs of FPR propellant were added to the S-IVB. For all cases in Figures 4 and 5, the LM/ATM flights have a positive payload weight margin, the minimum being 1200 lbs for the 56-day LM going into a 67° orbit. Since the SM SPS tanks are loaded to the "orbitable" capacity, the alternate mission is limited by the propulsive capability available on the CM/SM flights. This is clearly seen in Figure 4 where the apogee which can be reached decreases as the duration (spacecraft weight) increases.

It should be noted that the data shown in Figure 4 for the SPS suborbital ignition are not in all cases the ultimate performance achievable, although close to it. Time allotted for the study did not permit a full optimization with respect to the

SPS loading for the suborbital ignition. However, the performance is relatively insensitive to this parameter for values in the range of 15,000-20,000 lbs of consummable SPS propellant (Figure 6). The value of 15,000 lbs was used to generate the data shown in Figure 4. As seen in Figure 6, this is virtually optimum for higher yaw-rate cases, but a little conservative for the low-yaw cases.

Two specific circular-orbit cases were also run so that solar viewing opportunities from these orbits could be compared with the elliptic-orbit cases. Again the circular orbits indicated on Figure 4 are the highest which could be reached by using all the available SPS propellant to transfer the spacecraft from the parking orbit to the operational orbit, saving only enough propellant for the deorbit burn.

Once these results were available, the overall study concentrated on the following missions:

Duration (Days)	Inclination (Degrees)	Altitude (nm)	Suborbital SPS Ignition Required
56	50	150 x 400	Yes
28	50	150 x 400	No
28	50	150 x 600	Yes
28	50	375 x 375	Yes
28	63.5	150 x 300	Yes
28	63.5	225 x 225	Yes

The interest in 63.5° orbits stems from the fact that the apsidal rotation is zero at this inclination. That is, the argument of perigee of such an orbit would remain fixed during the mission. Thus, if the apogee location were selected for optimal solar viewing, it would remain so fixed throughout the mission.

Operational Considerations

In addition to the range safety aspects mentioned above, several other operational issues must be considered when weighing the relative merits of these high-inclination missions compared with the baseline low inclinations.

Flying missions at orbital inclinations other than near 29° adds to the rendezvous complexity. This is reflected primarily in the amount of phasing necessary to bring the chase vehicle and target vehicle into the correct relative position to effect the rendezvous terminal phase. Because orbit plane changes incur a substantial performance penalty, the chase vehicle must be launched essentially into the target orbit plane. Figure 7 shows a typical high inclination rendezvous launch window. There is only a very brief window each day when the wedge angle (which is a measure of the required plane change) is near zero. Note that only one such opportunity exists daily since the southerly launches must be eliminated for range safety reasons. With the in-plane window so short, there is no flexibility available to adjust the launch time in order to insure an in-phase condition as well. Referring to Figure 7, we see that if the LM/ATM is launched on Day 1, then on Day 2 it will be about 260° ahead of the CM/SM when the in-plane window for the CM/SM launch occurs. (The CM/SM, being in a lower orbit, "catches up" with the target.) The time required to complete the phasing depends on what orbit is used by the CM/SM. For example, if the CM/SM waits in a 120 nm circular orbit, approximately 57 orbits would be required to make up the 260° phase angle incurred on the Day-2 launch of the CM/SM. This situation will improve through Day 4 and then recycle on the following day to a very large phase angle. Because of this phasing problem, the LM/ATM must be capable of loitering on orbit for several days until the CM/SM can be launched and rendezvoused.

Finally, the high-inclination mission poses a number of problems for the landing and recovery. A launch abort or abort from orbit could result in a high-latitude landing where weather conditions and water temperatures have a higher probability of being beyond currently acceptable operational limits.

It must be recognized that the restriction of backup deorbit to occur at elliptical orbit apogee commits the landing after such a maneuver to the vicinity of perigee. If, for solar viewing optimization, the line of apsides of the operational orbit is aligned generally in the north-south direction, the backup deorbit landing would then occur near one of the latitude extremes of the orbit. Consideration has been given to the possibility of backup deorbit from positions other than apogee in order to move the landing point away from the perigee of the operational orbit. The penalty associated with this scheme can be inferred by considering the cost of shifting the perigee of the transfer ellipse achieved after the deorbit burn. Figure 8 shows the Δv required to shift the transfer orbit perigee a given amount by making the deorbit burn at the position in the operational orbit optimized for minimum Δv. It is seen that moving this perigee any significant amount is relatively expensive compared with the deorbit at apogee. In addition, because of the spherical geometry of earth orbits, the latitude of the landing site is reduced only a fraction of the angular shift of transfer-orbit perigee. The actual amount of this latitude shift depends on the direction of the line of apsides of the operational orbit and the orbit inclination.

Conclusion

It appears that, from the point of view of launch vehicle and spacecraft propulsion performance, high inclination CM-SM/LM-ATM missions are feasible in either elliptical or circular orbits. The altitude/inclination capability for such missions has been calculated. Several new operational problems arise in conjunction with these high-inclination missions such as significantly more phasing time required for rendezvous and the potential of high-latitude landing and recovery.

The relative merits of these orbits for solar viewing and ATM scientific return, as well as other operational aspects of the mission, are discussed in companion memoranda on the overall study.

I. Hirsch

K. E. Martersteck

1025-$\frac{IH}{KEM}$-dcs

Reference

1. "Solar Viewing Capability in High Inclination Circular and Elliptical Orbits - Case 620", Bellcomm Memorandum for File by B. D. Elrod, to be published.

2. "Earth Orbital Lifetime Prediction Model and Program", NASA TM X-53385 by A. R. Mc Nair and G. P. Boyken, February 1, 1966.

3. "Modifications to the BCMASP Simulator for Saturn IB Trajectories - Case 610", Bellcomm Memorandum for File by I. Hirsch, March 29, 1968.

4. "Spacecraft Weight Summary for CM-SM/LM-ATM Backup Mission - Case 620", Bellcomm Memorandum for File by W. W. Hough, August 13, 1968.

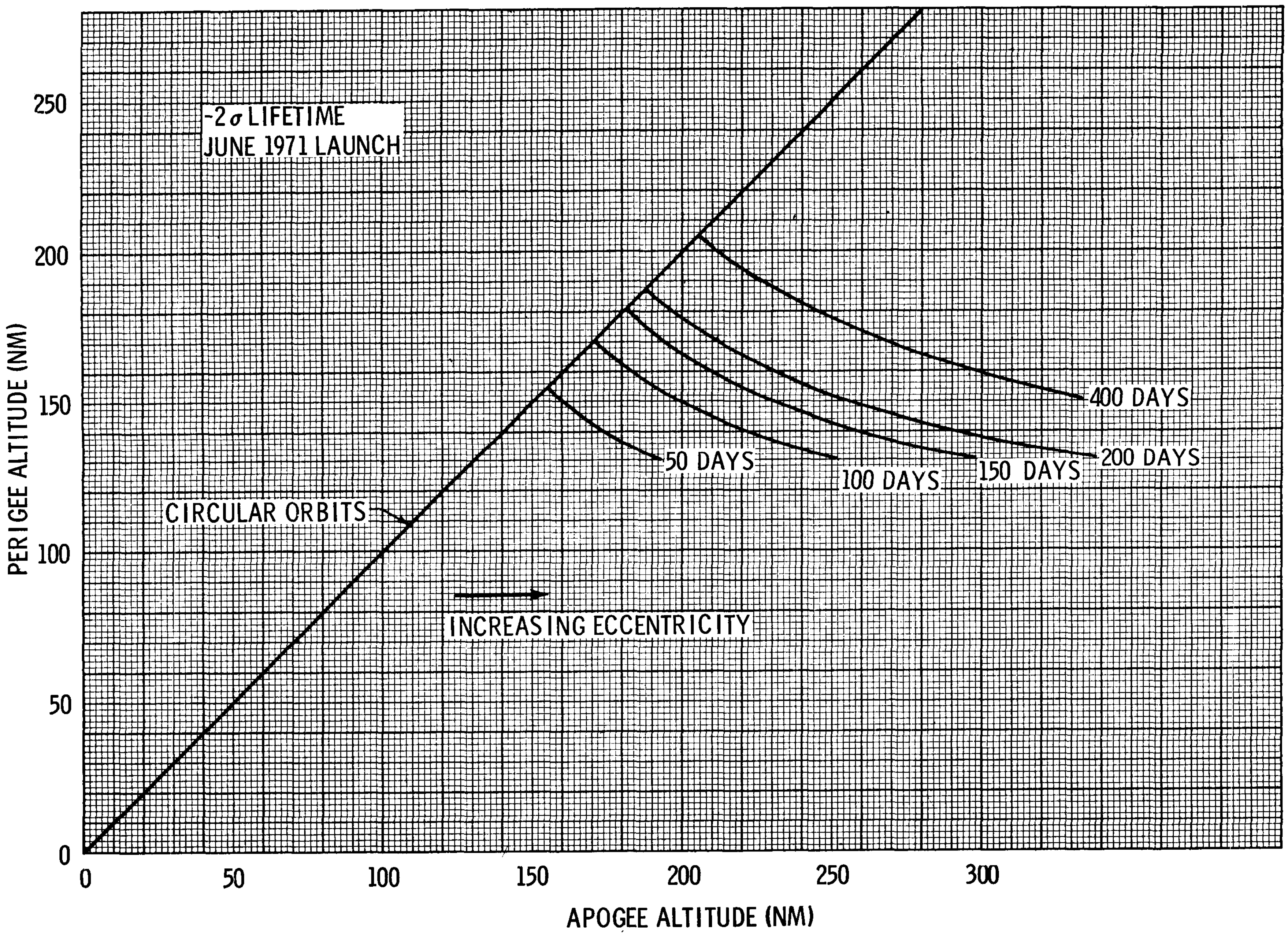

FIGURE 1 - ORBIT LIFETIME VS. INITIAL APOGEE & PERIGEE ALTITUDE

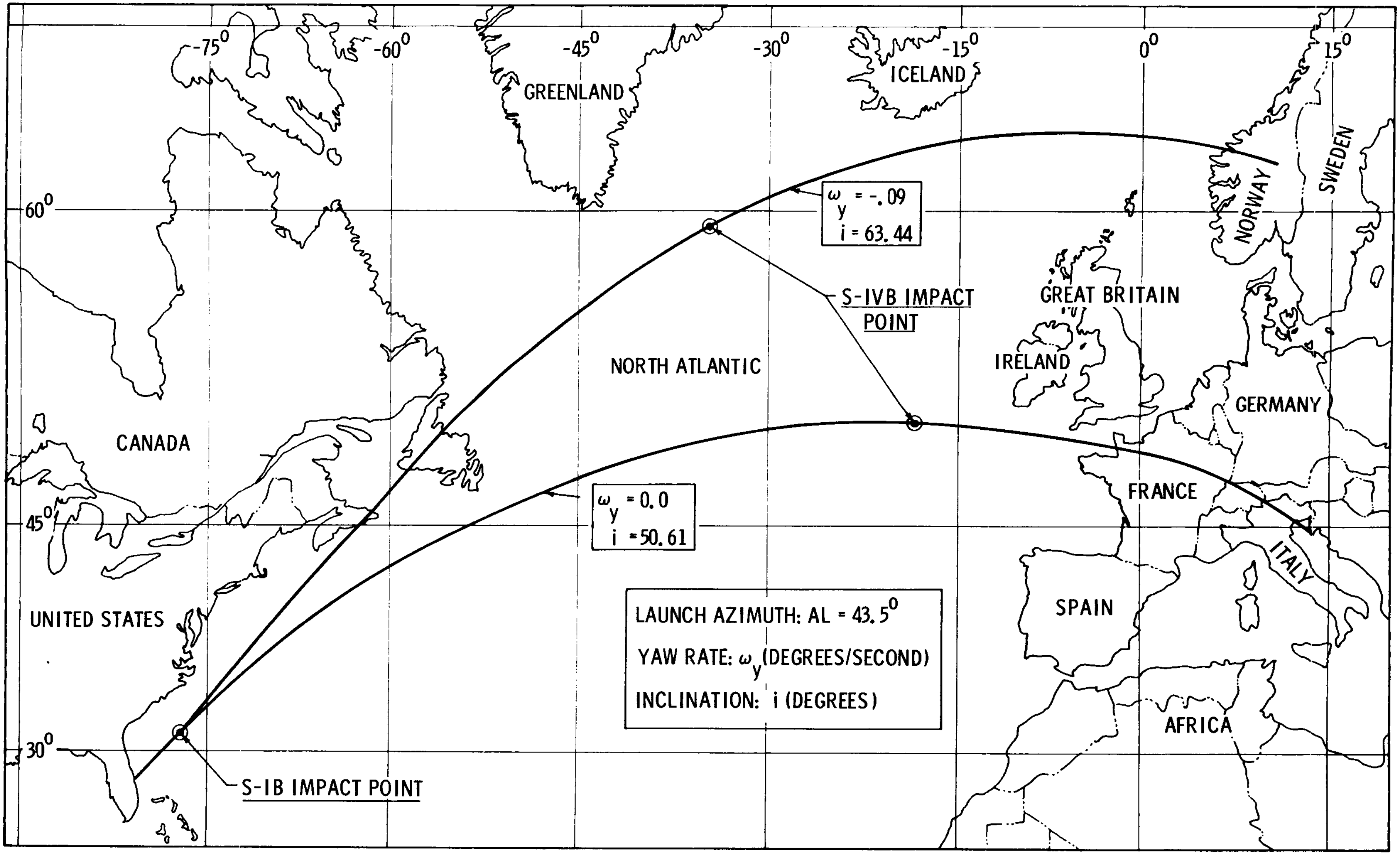

FIGURE 2 - INSTANTANEOUS IMPACT POINTS FOR HIGH INCLINATION MISSIONS

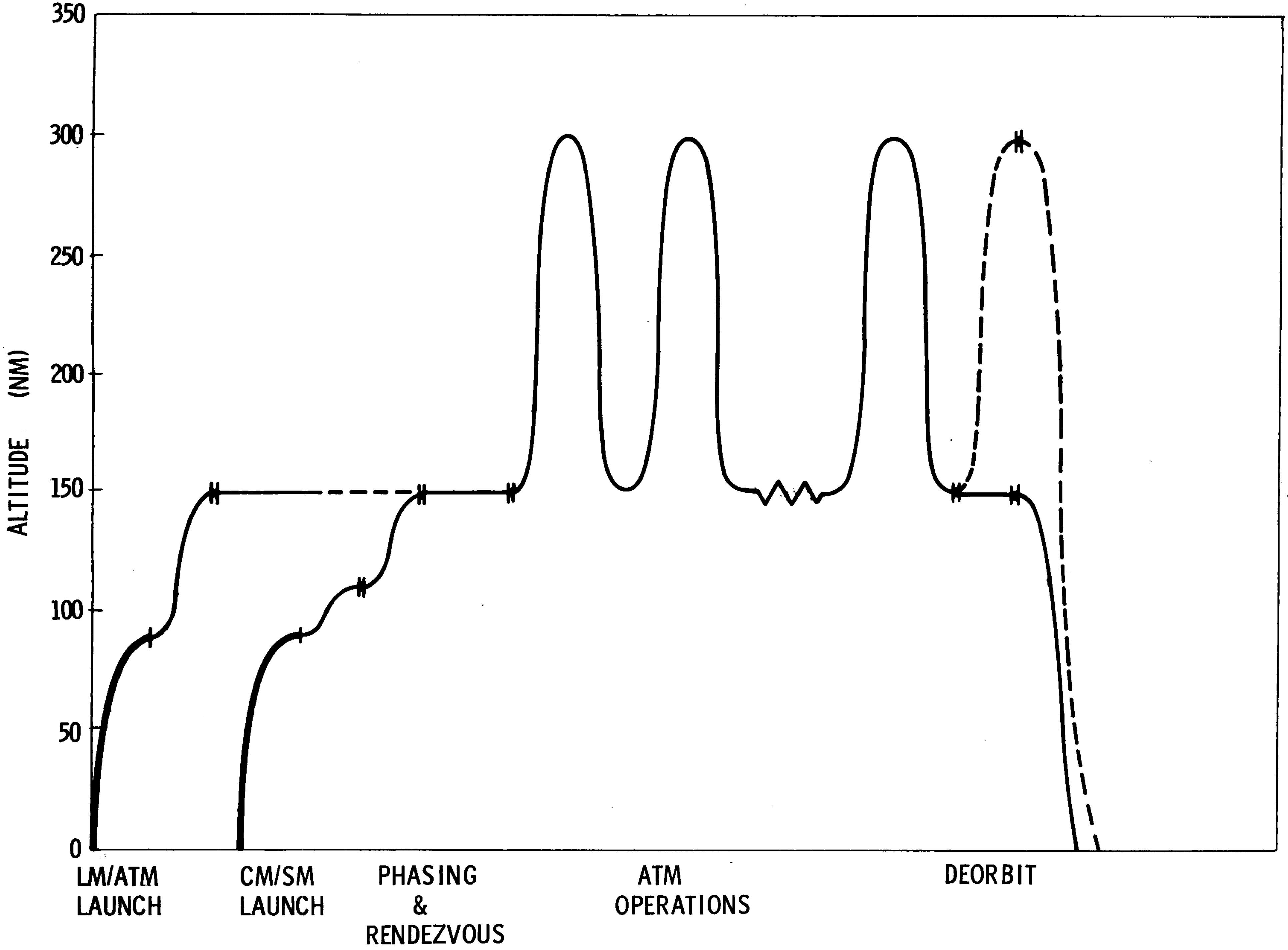

FIGURE 3 - LM/ATM ALTERNATE MISSION PROFILE

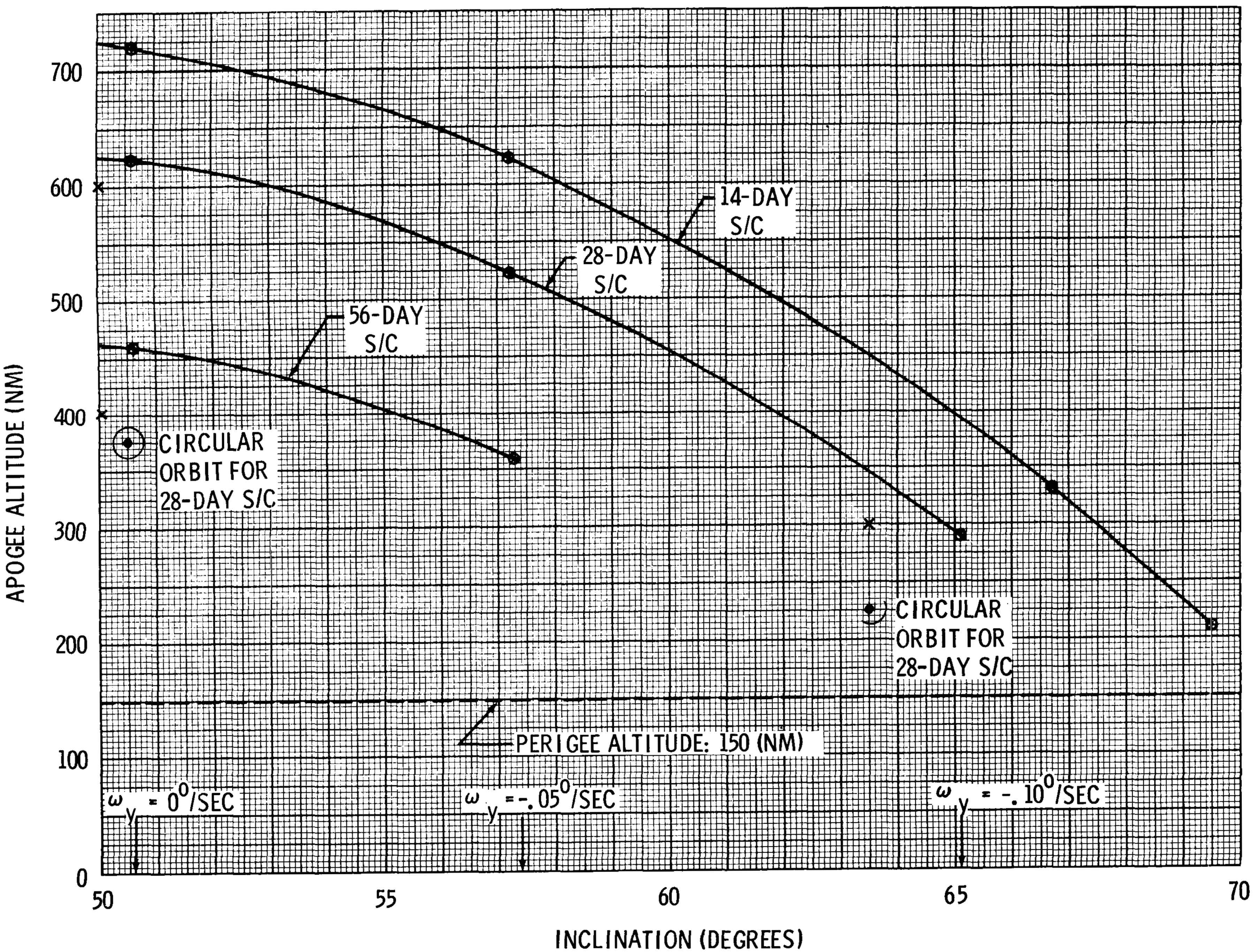

FIGURE 4 - ALTITUDE VS. INCLINATION FOR CM/SM-LM/ATM MISSIONS WITH SPS SUBORBITAL IGNITION

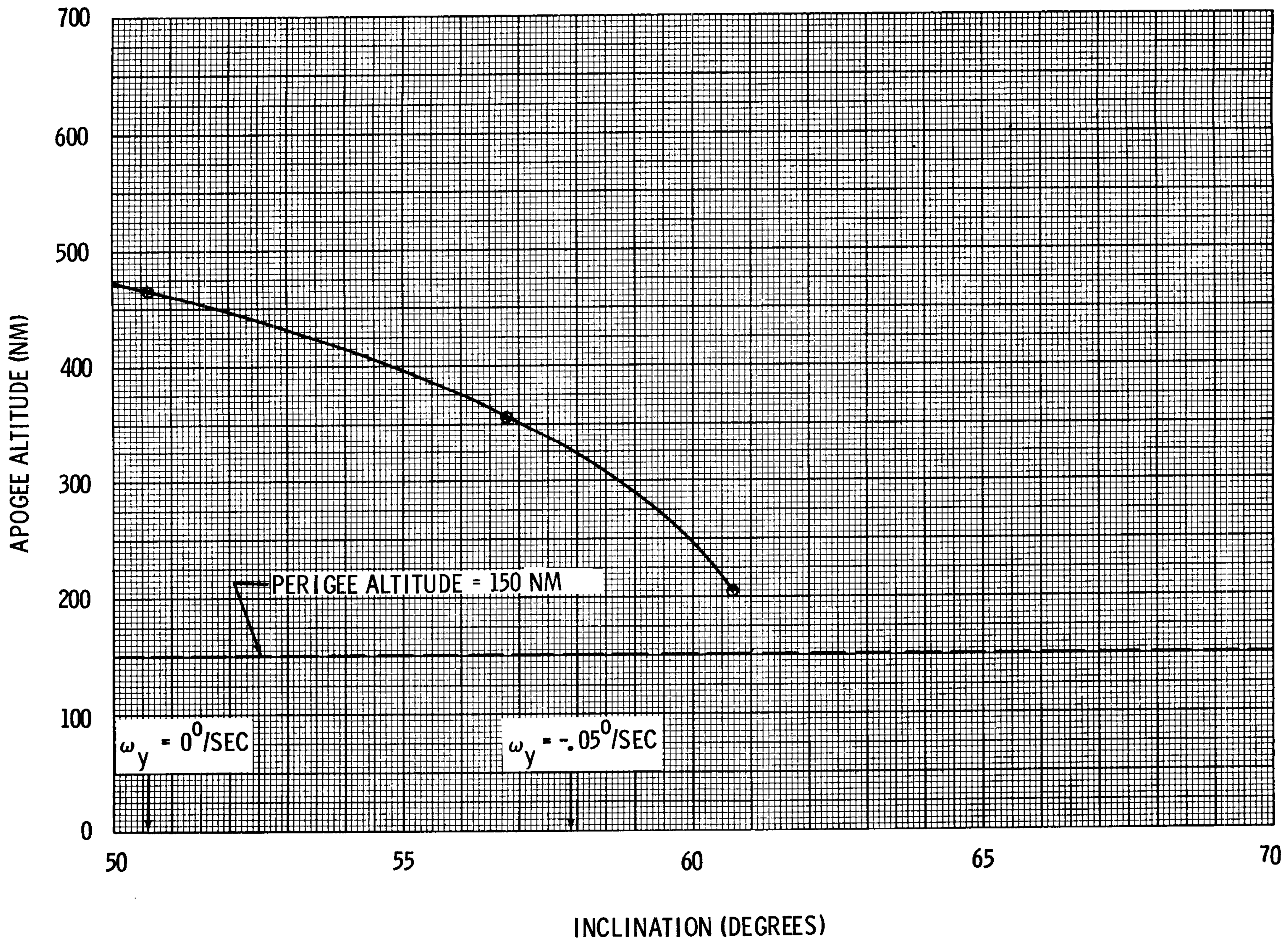

FIGURE 5 - ALTITUDE VS. INCLINATION FOR 28-DAY MISSION (2 STAGES TO ORBIT)

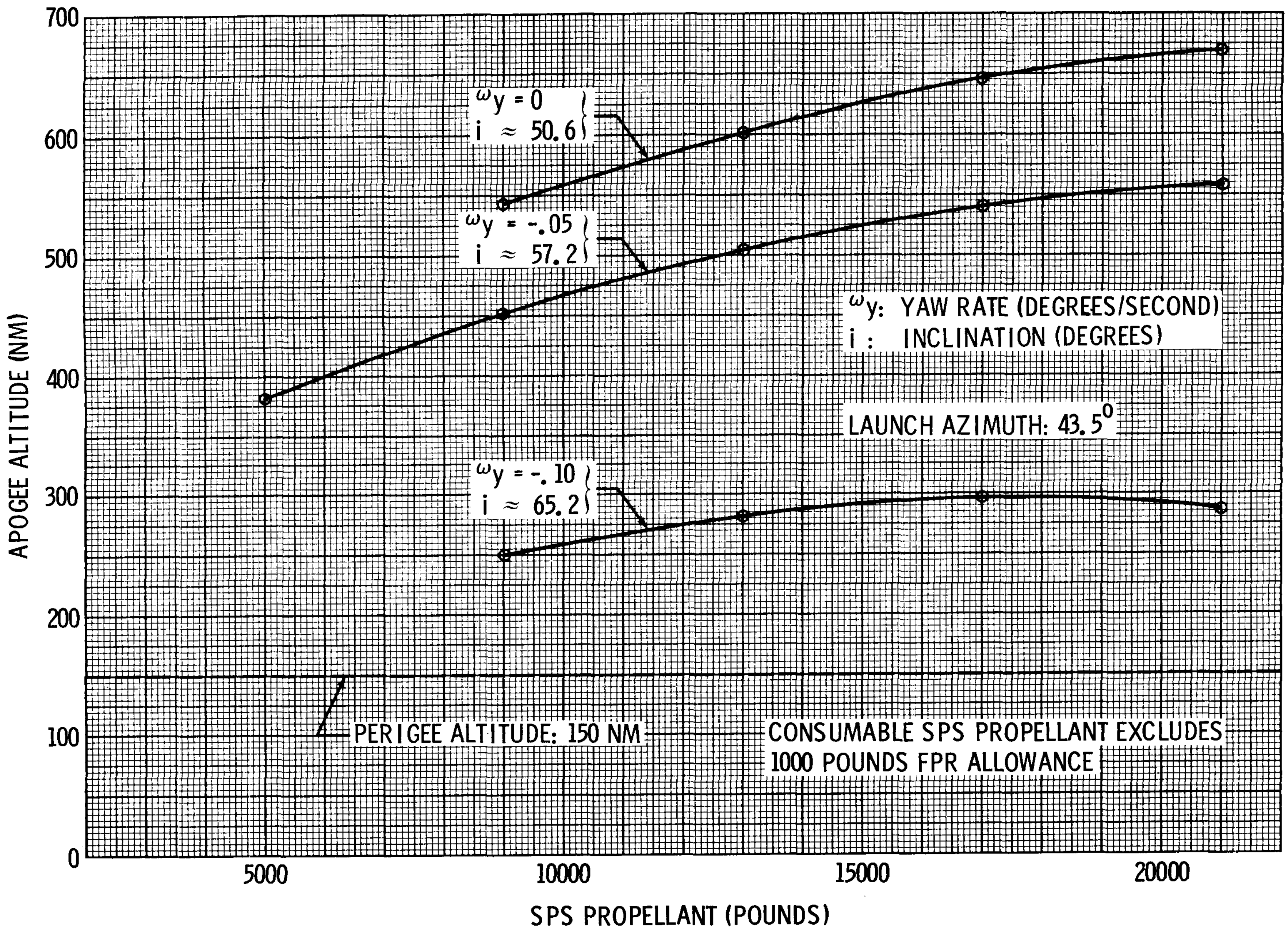

FIGURE 6 - APOGEE ALTITUDE VS. CONSUMABLE SPS PROPELLANT FOR 28-DAY MISSION

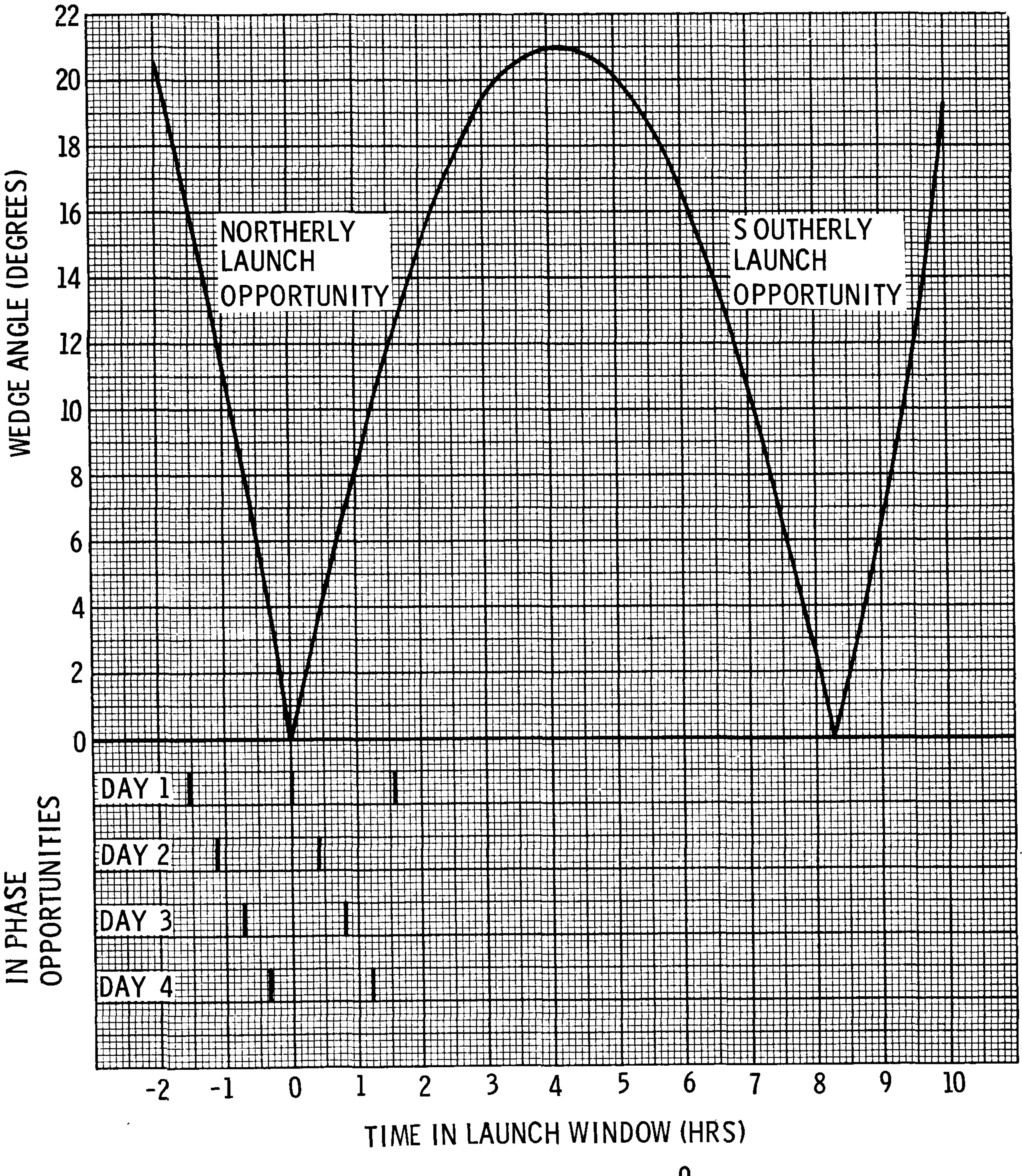

FIGURE 7 - RENDEZVOUS LAUNCH WINDOW FOR 50^{0} INCLINATION, 150 x 150 NM TARGET ORBIT

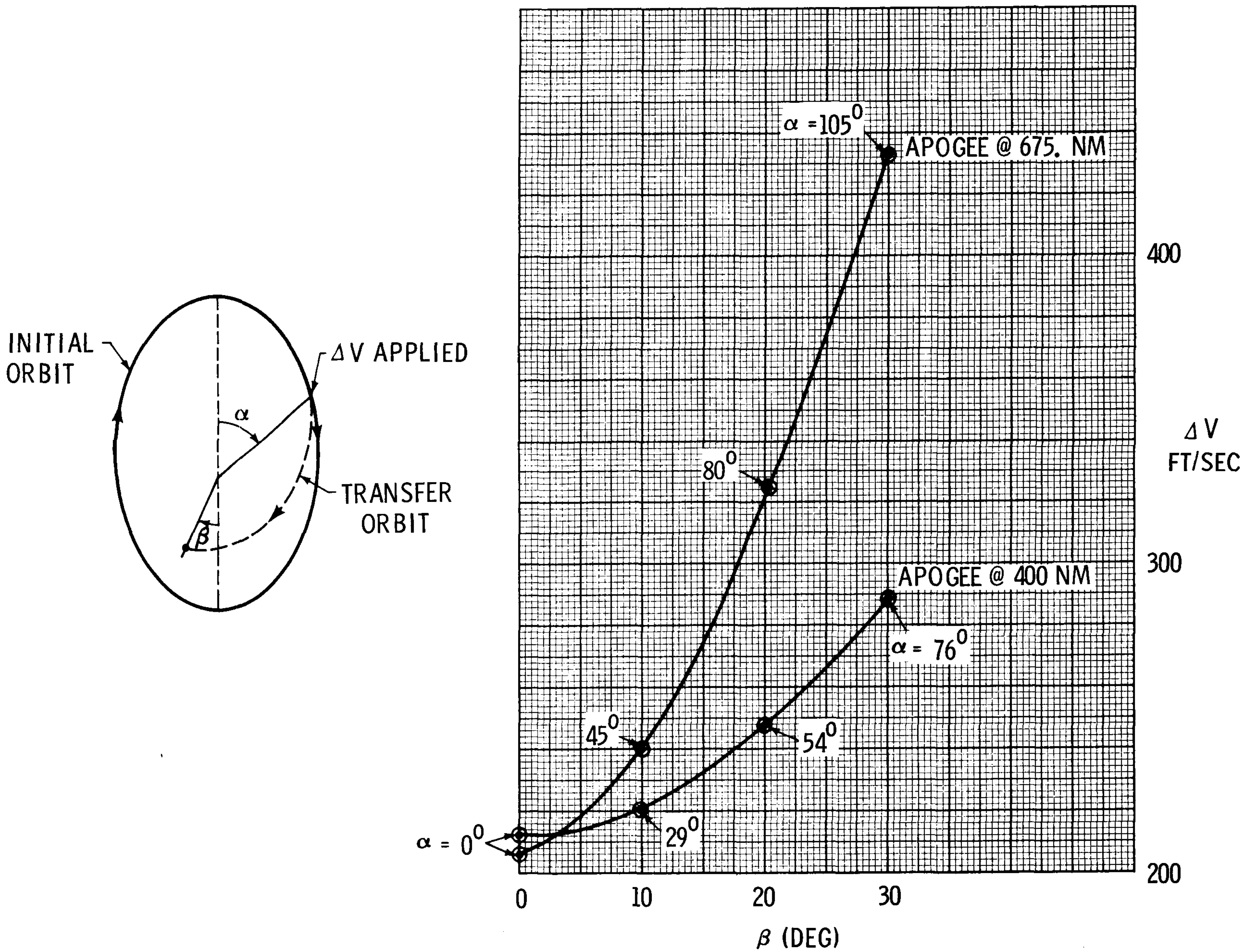

FIGURE 8 - MINIMUM ΔV REQUIRED TO LOWER PERIGEE FROM 150 TO 30 NM AND SHIFT IT BY β DEG.